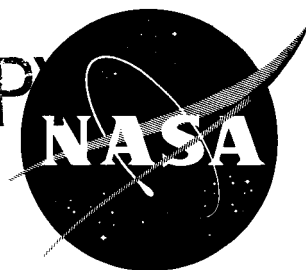


EXTRA COPY



Copy 1

TECHNICAL NOTE

D-635

FATIGUE INVESTIGATION OF FULL-SCALE WING PANELS
OF 7075 ALUMINUM ALLOY

By Claude B. Castle and John F. Ward

Langley Research Center
Langley Field, Va.

LIBRARY COPY

APR 26 1961

SPACE FLIGHT
LANGLEY FIELD, VIRGINIA

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON

April 1961

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL NOTE D-635

FATIGUE INVESTIGATION OF FULL-SCALE WING PANELS
OF 7075 ALUMINUM ALLOY

By Claude B. Castle and John F. Ward

SUMMARY

Results are presented of a fatigue investigation conducted using 18 outer wing panels of T-29A airplanes. Constant-amplitude tests were performed using the resonant-frequency method at three different alternating load levels superposed on a 1g, or level-flight mean load. Information is presented concerning crack location, cycles to crack initiation, crack propagation, and residual static strength.

INTRODUCTION

The ultimate goal of the fatigue analysis of an aircraft structure is the determination of a reliable safe service life for the airframe. The fatigue analysis should be based on a sufficient backlog of data so that it would be unnecessary to resort to extremely high factors of safety on calculated service life. In determining the safe service life of an aircraft, it is necessary to have basic data concerning load-lifetime relationships and associated scatter limits, crack propagation characteristics, and the residual static strength of the structure following the propagation of fatigue cracks. This information must be applicable to the specific type of material and type of construction for the particular structure under consideration.

As part of a research program initiated to investigate the fatigue characteristics of various types of full-scale aluminum alloy built-up structures, fatigue tests were conducted using nine pairs of 7075 aluminum alloy outer wing panels from T-29A transport airplanes. The present report summarizes the results of constant-amplitude-loading fatigue tests and residual-static-strength tests for the nine pairs of wing panels.

EQUIPMENT AND PROCEDURE

Test Specimen

The specimens used in the tests were the outer wing panels from the T-29A military version of the CV 240 (Convair) airplane. The wing panels had become obsolete, as a result of an extensive design modification, and had been removed from in-service aircraft. The panels had accumulated between 662 and 925 flight hours (average 725 hours) prior to salvage.

Some characteristics of the wing as well as other pertinent data for the test aircraft are as follow:

Design gross weight, lb	39,000
Level-flight equivalent air speed, V_c , mph	285
Wing area, sq ft	817
Wing span, ft	92.1
Mean aerodynamic chord, in.	117
Aspect ratio	10
Design ultimate load factor	4.7

The test specimen consisted of that portion of the wing structure from a chordwise wing splice at rib station 16 to rib station 26 as indicated by the shaded area on the airplane planform shown in figure 1(a). The specimens were 212 inches in length and consisted of the wing-spar box and the trailing-edge structure from the rear spar rearward to the aileron hinge brackets, including the upper and lower thin gage skin and access panels.

In flight the lower surface of the test panel is subjected to steady and alternating tension loadings and therefore is the critical surface with regard to fatigue damage. The test panel lower or tension surface layout is shown in figure 1(b). The panel was of all-metal, riveted, stressed-skin construction. The tension surface from rib stations 16 to 20 consisted of skin and extruded stiffeners of 7075-T6 aluminum-alloy material. From rib stations 20 to 26, the skin was of 2024-T81 clad material and the stiffeners were of 2024-T84 clad material. There were skin splices located chordwise at rib stations 20 and 22 and a spanwise splice along stiffener 7. The spar caps were of 7075-T6 extrusions and the spar webs were fabricated from 7075-T6 clad material. The distribution of tension-surface cross-sectional area at rib station 17 is shown in figure 1(c).

The chordwise production splice at the inboard end of the test panel, rib station 16, consisted of attach angles along the full-tension

and compression surfaces and both spar webs. At the inboard end of each stiffener and spar cap, there was a heavy fitting through which the stiffener and spar-cap loads were transferred into the panel attachment bolts.

There were numerous access cutouts in the tension surface. The largest of these was a square cutout with a load-carrying cover plate located between rib stations 16 and 17. This cutout was used for access to the wing-panel attachment bolts at rib station 16. The large cover plate was in place during all tests. There was a round cutout in the center of the large cover plate which enabled visual inspection of the wing attach fittings without removing the large cover plate. Eight access cutouts were located spanwise between stiffeners 6 and 7. The cover plates on these cutouts were also of the load-carrying type and were in place during all tests. At rib station 16, a small cutout was located over both the front- and rear-spar end fittings. These cutouts provided access to the large attachment bolts in the spar-cap end fittings.

Loading Apparatus

The loading apparatus was designed to simulate the flight load distribution on the test panel. The test loads consisted of a steady mean loading associated with 1g level flight, together with each of three constant-amplitude alternating loadings. Duplication of the design bending-moment distribution was achieved in the test setup through the use of deadweights attached to the front and rear spars at seven spanwise stations on each panel. The wing-torsion distribution for 1g flight was achieved by selecting the proper proportion of deadweight attached at the front and rear spars at each spanwise station.

The general arrangement of the fatigue machine and specimens is shown in figure 2. The left and right specimens, with the tension surface facing upward, were attached to steel center sections mounted on a structural steel backstop. The alternating loads were applied dynamically by driving the wing panels and rigidly attached deadweights at the natural bending frequency of 400 cpm. The 1g mean loading was achieved by suspending additional deadweights from soft shock cord so that the sum of the rigidly attached and suspended deadweights equalled the 1g mean-load distribution.

The wing was driven dynamically through push rods and eccentrics at the wing tips. This induced an inertia loading with the proper spanwise and chordwise distribution. The dynamic loads were applied by bringing the drive system up to 400 cpm at zero eccentricity and then introducing a preset eccentricity at the push rod. It was necessary to have both left and right panels in place during the fatigue tests.

By loading the panels symmetrically, it was possible to eliminate back-stop rotation which would have upset the dynamic-loading distribution. A comparison of the 1 g design bending moment and the calculated applied 1 g bending moment is shown in figure 3. The push rods at the wing tips contained shear pins to prevent inadvertent overloading which would alter the desired bending-moment distribution.

In preparation for the residual-static-strength test after completion of the fatigue test, the deadweights were removed and a whiffle-tree assembly was attached to the wing panel. The wing panel was then loaded by means of hydraulic rams until complete static failure of the tension surface occurred.

Test Procedures

Three pairs of wing panels were tested at each of three constant-amplitude load levels with alternating load factors Δn of $\pm 0.25g$, $\pm 0.42g$, and $\pm 0.76g$. These alternating load factors used in the tests corresponded to gust intensities of ± 4.8 fps, ± 8.0 fps, and ± 14.5 fps, respectively. All of these tests were run with a 1 g mean load. The magnitudes of the steady and dynamic applied bending moments were monitored at five spanwise stations with the use of strain-gage bridges.

As each pair of test panels was subjected to cyclic loading, frequent periodic inspections were made to detect the initiation of any fatigue cracks and to record the subsequent crack growth. The early detection of fatigue cracks in inaccessible areas was made possible through the use of bonded wire crack-detector circuits. The use of these bonded wires to detect small fatigue cracks is described in appendix A of reference 1. Once a crack was found, a detailed record of its growth was kept, and each measurement was correlated with the number of cycles of load applied. Visual inspection of internal elements was accomplished through access cutouts with the aid of a hand mirror. X-ray inspection of the spar caps was also performed at frequent intervals.

The fatigue cracks grew to different extents in various specimens. Due to the fact that the wing panels were tested in pairs and the cracks propagated at different rates, the test was terminated when a high percentage of tension-surface area was lost due to crack propagation in one of the panels. At the completion of the fatigue test, each panel was statically loaded to destruction to determine the residual static strength of the tension surface containing fatigue cracks.

RESULTS AND DISCUSSION

Fatigue Tests

For identification purposes the fatigue cracks observed were listed according to their general location, and a code letter was assigned to each area. Table I lists the code letters and respective location for each type of crack observed. The general locations are also indicated in figure 4. Figure 4(a) shows the location of spar and stiffener cracks, and figure 4(b) shows the location of skin cracks. When a crack was included in the path of the static failure of a given specimen, it was considered a critical crack.

A complete listing of all observed cracks is presented in table II, which lists specimen number, crack number, type, location, and number of cycles to initiation. In regard to life-to-crack initiation, cracks were generally first observed when they were about 1/4 inch in length. In the instances when this was not the case and considerable growth had taken place, lifetime to initiation was estimated from crack-propagation data for cracks of a similar type under the same loading conditions. The specimen numbers shown in table II were assigned to indicate the order of testing only and do not imply left and right wing panels taken from the same aircraft.

Crack locations.- All fatigue cracks occurred at the inboard end of the test panel between rib stations 16 and 20. This area contained many structural discontinuities such as the large access cutout and the stiffener and spar end-attachment fittings.

The stiffener cracks A occurred in stiffeners 5 and 11 and originated in a sharp radius shown in figure 5(a), where a portion of the upper horizontal flange of the stiffener had been removed to provide clearance for inserting the stiffener end-fitting attachment bolts. These cracks initiated and grew to a rivet hole in the vertical web of the stiffener and terminated there. Cracks B (fig. 5(a)) originated in the vertical web of the stiffeners at the most outboard rivet hole of the stiffener end fittings. These outboard-end rivet holes were the most highly loaded, and cracks which originated in this location grew through the entire stiffener. Stiffener cracks C initiated at rivet holes in the vertical web due to overloading of the stiffener as a result of a skin crack propagating across the stiffener.

Spar cracks D originated in the most outboard rivet hole of the spar and spar end fitting as shown in figure 5(b). The cracks initiated at this point because it is the minimum cross section carrying the total force in the spar, and the first rivet hole in the end fitting occurs here. It is well known that the first connector in a line of connectors

produces the highest concentration of force. Crack E (fig. 5(b)) originated in the spar cap at a rivet hole just inboard of the juncture between the spar and rib station 18. Spar crack F (fig. 5(b)) occurred in the spar end fitting and was the result of overloading of the fitting at rib station 16 due to skin and stiffener failures in the vicinity.

Skin cracks G (fig. 4(b)) initiated as a direct result of the presence of spar cracks D and E. These spar cracks initiated and began to propagate, which in turn overloaded the skin at the respective rivet hole and induced the initiation of the skin cracks. Skin cracks H originated at the outboard corner of the large access cutout (fig. 4(b)). These cracks initiated and grew slowly until affected by the presence of other cracks in the vicinity. Crack I initiated at a rivet hole over stiffener 10. This rivet hole was at the outboard end of a doubler over stiffener 10. The propagation of skin crack I caused overloading and subsequent failure of stiffener 10. Skin cracks J originated at rivet holes over stiffener 6 adjacent to the small access cutouts between rib stations 18 and 20. Shear web crack K occurred only once and originated in a front-spar shear web lightening hole in the vicinity of rib station 17.

Cracks L and M, which occurred in the skin and doubler material and the large cover plate, respectively, just outboard of rib station 16, initiated after local overloading caused by the propagation of stiffener cracks B.

Summary of crack occurrences.- A summary of crack occurrences is presented in table III. The numbers appearing in the table indicate the order of occurrence of each crack in any given test specimen. There were as many as 15 cracks per specimen. A trend toward increased number of crack occurrences per panel at the high load levels is also indicated. The total number of occurrences for any given crack and the number of times the crack was critical are also summarized in table III. The spar-cap cracks were present in every specimen and were almost always critical.

Cycles to crack initiation.- The number of cycles to initiation of each crack, as listed in table II, is plotted against load level in figure 6. This figure is subdivided into three parts, one each for stiffener, spar, and skin cracks. In order to avoid excessive overlapping, some of the test-point symbols are plotted above and below the test load level indicated by the horizontal dashed line. Considerable scatter is evident in figure 6 for all cracks at all load levels. There was no correlation between previous flight time and number of cycles to crack initiation.

Crack propagation.- Crack propagation was characterized by the initiation and subsequent growth of cracks in many locations. Cracks

developed until a significant portion of the tension surface had failed. In general, the pattern of cracking was not consistent from specimen to specimen at any load level. However, it was usually necessary for a spar crack to be present before crack propagation developed to a significant extent. This was due to the fact that the spar caps represented a substantial portion of the cross-sectional area of the tension surface as indicated in figure 1(c).

Crack-propagation curves are shown in figure 7 for each specimen tested. A comparison of the type of propagation curve in figure 7 and the total number of crack occurrences from table III indicates that for curves with steep slopes (for example, L-2 and R-2 in fig. 7(b)), the total number of cracks present was low, and spar cracks were always included and were among the first to initiate. For the propagation curves characterized by shallow slopes, the number of cracks present was high (for example L-3 and R-3 in fig. 7(b)) and spar cracks were among the last to initiate. In general, if spar cracks initiated first in the order of crack occurrence, crack propagation to 20 or 30 percent of the tension area was accomplished rapidly. If spar cracks occurred late in the order of occurrence, the many stiffener and skin cracks contributed somewhat to the tension area failed but significant loss of area did not take place until the spar cracks propagated.

Load-lifetime comparison.- In figure 8 the lifetime to final failure of the T-29A outer wing panels is compared with the envelope of the load-lifetime data presented in figure 13 of reference 1. The envelope represents the limits of final-failure lifetime data from full-scale fatigue tests of five types of aircraft. Final failure lifetime for the T-29A data was considered as the total number of cycles for failure of 25 percent of the tension surface area. This lifetime was estimated from the crack-propagation data. The T-29A data were corrected to a load ratio of zero in the manner outlined in reference 1. As indicated in figure 8, the T-29A final-failure lifetime data compare favorably with the summary of other full-scale load-lifetime data presented in reference 1.

Residual-Static-Strength Tests

Following the cyclic-loading tests, in which cracks were allowed to grow until 16 to 50 percent of the tension surface area had failed due to crack propagation, the panels were statically loaded to failure. A typical result of a static-loading test is shown in figure 9. The panel shown in this figure is L-6 which was run at a load level of $\pm 0.76g$ until crack propagation had caused 30 percent of the tension surface to fail.

The pattern of static failure was difficult to predict in detail. This was because the determination of all the fatigue cracks that would be involved in the static failure could not be established due to the complex path of the static failure. After static failure, an accurate determination of the tension surface area lost to fatigue cracking was made by considering only those cracks actually included in the static-failure path. This information is presented in figure 10 plotted against residual static strength, which is presented as a percent of the calculated ultimate strength of the undamaged tension surface. These data again exhibit large scatter due to the variety of the failure patterns. An average value of the static-strength reduction factor, K , of 1.55 is indicated by the dashed line in figure 10. Scatter about this line is felt to be partly the result of the fact that various crack patterns existed for any given value of percent of tension area failed, resulting in a wide variation in residual-static-strength characteristics.

CONCLUSIONS

The following conclusions are based on the results of the fatigue tests and residual-static-strength tests of the outer wing panels of T-29A transport airplanes:

1. Cracks initiated in as many as 15 locations per test specimen.
2. The average number of cracks occurring in any given specimen was the greatest at the highest load level, $\pm 0.76g$.
3. Lifetime to initiation of cracks was characterized by a large amount of scatter.
4. Spar-cap cracks were necessary to achieve a significant loss in percentage of tension surface area. In general, if spar cracks initiated early in the order of crack occurrence, crack propagation to 20 or 30 percent of the tension area was rapid. If spar cracks initiated late in the order of crack occurrence, the many stiffener and skin cracks contributed somewhat to the tension area failed, but significant loss of area did not take place until the spar cracks propagated.
5. The pattern of crack propagation was not consistent and this was believed to be one cause of the considerable spread in the residual static-strength results.

6. The final-failure lifetime data for the T-29A airplane wing panels compare favorably with the summary of other full-scale load-lifetime data presented in NACA TN 3190.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., September 21, 1960.

REFERENCE

1. McGuigan, M. J., Jr., Bryan, D. F., and Whaley, R. E.: Fatigue Investigation of Full-Scale Transport-Airplane Wings - Summary of Constant-Amplitude Test Through 1953. NACA TN 3190, 1954.

TABLE I.- CRACK LOCATIONS

Element	Code letter	Location	Remarks
Stiffeners	A	Cutout radius in stiffener 5 or 11 flanges just outboard of rib station 16	Cracks occurred early in the test and did not propagate throughout the test
	B	Outboard rivet hole at end fitting attachment in stiffener 5, 6, 10, or 11	Same as A
	C	Stiffener 5, 6, or 10 between stations 17 and 18	Cracks were the result of local overloading of the stiffener due to the presence of a skin crack
Spars	D	Outboard rivet hole at front or rear spar end fitting attachment	
	E	Rear spar, 25.5 in. outboard of station 16	
	F	Rear spar end fitting, 2 in. outboard of station 16	
Skin	G	Rivet hole over front or rear spar fatigue crack	Cracks were a direct result of the spar failures D and E
	H	Outboard corners of large cutout	Cracks initiated and grew slowly until affected by other cracks propagating in the vicinity.
	I	Over stiffener no. 10, 22 in. outboard of station 16	Same as H
	J	Rivet hole next to small access cutouts	Never exceeded 1 inch in length
Shear web	K	Front spar shear web	Caused by local buckling of shear web at high load level
Skin and doubler combination	L	Rivet hole 2 in. outboard of station 16 next to rear spar end fitting	
Coverplate	M	Holes in large coverplate	Loss in area at inboard end of coverplate caused overloading in adjacent skin and stiffeners, which in turn caused accelerated crack propagation in these members

TABLE II.- SUMMARY OF OBSERVED CRACKS

(a) Load level, Δn , $\pm 0.25g$

Specimen number	Crack number	Type	Cycles to crack initiation	Location
L-7	1	D	1,555,000	Outboard rivet hole in rear-spar end fitting
	2	B	1,670,000	Outboard rivet hole in end fitting of stiffener 11
	3	G	2,250,000	Outboard rivet hole at rear-spar end fitting
	4	C	2,760,000	Stiffener 5, 16 in. outboard of station 16
	5	H	2,860,000	Outboard rearward corner of large cutout
	6	C	3,268,000	Stiffener 6, 17 in. outboard of station 16
R-7	1	A	300,000	Cutout in stiffener 5
	2	B	1,000,000	Outboard rivet hole in end fitting of stiffener 5
	3	J	1,280,000	Rivet hole next to first small cutout
	4	L	2,400,000	Rivet hole 2 in. outboard of station 16, next to rear-spar end fitting
	5	F	3,500,000	Rear-spar end fitting, 2 in. outboard of station 16
	6	C	9,100,000	Stiffener 5, 25.5 in. outboard of station 16
	7	C	9,200,000	Stiffener 6, 25.5 in. outboard of station 16
	8	B	11,973,000	Outboard rivet hole in end fitting of stiffener 6
L-8	1	J	1,130,000	Rivet hole next to first small cutout
	2	D	1,700,000	Outboard rivet hole in rear-spar end fitting
	3	C	1,900,000	Stiffener 5, 16 in. outboard of station 16
	4	G	3,470,000	Second rivet hole inboard of outboard end of rear-spar end fitting
	5	G	3,830,000	Outboard rivet hole at rear-spar end fitting
	6	C	4,000,000	Stiffener 6, 17 in. outboard of station 16
R-8	1	A	200,000	Cutout on stiffener 5
	2	H	390,000	Outboard forward corner of large cutout
	3	B	1,100,000	Outboard rivet hole in end fitting of stiffener 6
	4	L	1,510,000	Rivet hole 2 in. outboard of station 16, next to rear-spar end fitting
	5	D	4,400,000	Outboard rivet hole in front-spar end fitting
	6	G	4,800,000	Outboard rivet hole at front-spar end fitting
	7	F	5,490,000	Rear-spar end fitting, 2 in. outboard of station 16
	8	B	6,300,000	Outboard rivet hole in end fitting of stiffener 10
L-9	1	D	5,085,000	Outboard rivet hole in rear-spar end fitting
	2	D	5,140,000	Outboard rivet hole in front-spar end fitting
	3	G	5,280,000	Outboard rivet hole at rear-spar end fitting
	4	G	5,340,000	Outboard rivet hole at front-spar end fitting
R-9	1	J	1,590,000	Rivet hole next to first small cutout
	2	E	1,628,000	Rear-spar rivet hole, 25.5 in. outboard of station 16
	3	G	1,670,000	Over-rear-spar rivet hole, 25.5 in. outboard of station 16
	4	C	1,750,000	Stiffener 5, 25.5 in. outboard of station 16
	5	C	1,830,000	Stiffener 6, 17 in. outboard of station 16

TABLE II.- SUMMARY OF OBSERVED CRACKS - Continued

(b) Load level, Δn , $\pm 0.42g$

Specimen number	Crack number	Type	Cycles to crack initiation	Location
L-1	1	D	545,000	Outboard rivet hole in front-spar end fitting
	2	G	574,000	Outboard rivet hole at front-spar end fitting
R-1	1	A	155,000	Cutout in stiffener 5
	2	B	240,000	Outboard rivet hole in end fitting of stiffener 6
	3	G	572,000	Outboard rivet hole at rear-spar end fitting
	4	D	580,000	Outboard rivet hole in rear-spar end fitting
L-2	1	E	200,000	Rivet hole 25.5 in. outboard of station 16 in rear spar
	2	G	210,000	Skin rivet hole 25.5 in. outboard of station 16 over rear spar
	3	D	255,000	Outboard rivet hole in front-spar end fitting
	4	C	255,000	Stiffener 5, 25.5 in. outboard of station 16
R-2	1	F	140,000	Rivet hole, 2 in. outboard of station 16, in the rear-spar end fitting
	2	B	155,000	Outboard rivet hole in end fitting of stiffener 5
	3	D	215,000	Outboard rivet hole in rear-spar end fitting
	4	G	225,000	Skin rivet hole over outboard rear-spar end fitting
L-3	1	A	138,000	Cutout in stiffener 5
	2	H	145,000	Outboard forward corner of large cutout
	3	B	205,000	Outboard rivet hole in end fitting of stiffener 5
	4	I	282,000	Over stiffener 10, 22 in. outboard of station 16
	5	H	308,000	Outboard rearward corner of large cutout
	6	D	360,000	Outboard rivet hole in rear-spar end fitting
	7	B	415,000	Outboard rivet hole in end fitting of stiffener 10
	8	J	425,000	Rivet hole next to second small cutout
	9	J	425,000	Rivet hole next to first small cutout
	10	L	470,000	Rivet hole 2 in. outboard of station 16, next to rear-spar end fitting
	11	C	515,000	Stiffener 10, 22 in. outboard of station 16
	12	B	530,000	Outboard rivet hole in end fitting of stiffener 6
	13	G	550,000	Out of first rivet hole outboard of rear-spar end fitting
	14	A	565,000	Cutout in stiffener 11
R-3	1	A	160,000	Cutout in stiffener 5
	2	J	170,000	Rivet hole next to first small cutout
	3	B	205,000	Outboard rivet hole in end fitting of stiffener 11
	4	B	260,000	Outboard rivet hole in end fitting of stiffener 5
	5	J	450,000	Rivet hole next to second small cutout
	6	J	450,000	Rivet hole next to first small cutout
	7	L	480,000	Rivet hole 2 in. outboard of station 16, next to rear-spar end fitting
	8	B	495,000	Outboard rivet hole at end fitting of stiffener 6
	9	M	550,000	Screw hole in large cover plate at station 16
	10	B	591,000	Outboard rivet hole at end fitting of stiffener 10
	11	D	745,000	Outboard rivet hole at front-spar end fitting
	12	G	772,000	Outboard rivet hole at front-spar end fitting
	13	M	800,000	Screw hole in large cover plate at station 16
	14	G	825,000	Second rivet hole inboard of outboard end of rear-spar end fitting
	15	D	839,000	Outboard rivet hole in rear-spar end fitting

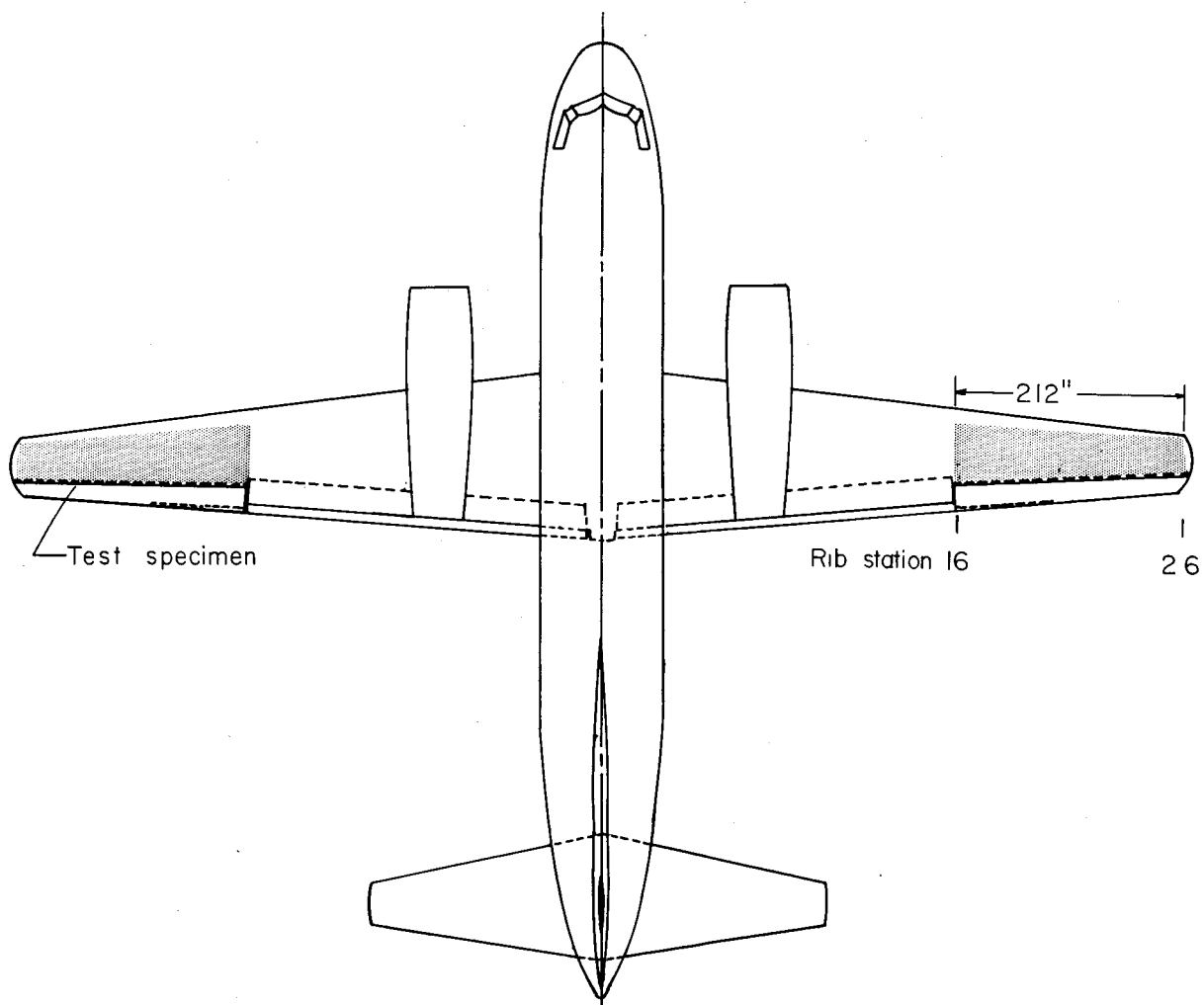
TABLE II.- SUMMARY OF OBSERVED CRACKS - Concluded

(c) Load level, Δn , $\pm 0.76g$

Specimen number	Crack number	Type	Cycles to crack initiation	Location
L-4	1	H	14,800	Outboard forward corner of large cutout
	2	H	18,400	Outboard rearward corner of large cutout
	3	A	19,000	Cutout in stiffener 5
	4	J	39,000	Rivet hole next to first small cutout
	5	B	51,000	Stiffener 10, 22 in. outboard of station 16
	6	B	57,500	Outboard rivet hole at end fitting of stiffener 10
	7	A	59,000	Cutout in stiffener 11
	8	B	64,000	Outboard rivet hole at end fitting of stiffener 6
	9	D	70,000	Outboard rivet hole at rear-spar end fitting
	10	M	82,000	Screw hole in large cover plate, station 16
	11	D	90,000	Outboard rivet hole at front-spar end fitting
	12	G	91,000	Outboard rivet hole at rear-spar end fitting
R-4	1	A	11,000	Cutout in stiffener 5
	2	B	37,200	Outboard rivet hole at end fitting of stiffener 11
	3	B	47,000	Outboard rivet hole at end fitting of stiffener 5
	4	J	49,400	Rivet hole next to first small cutout
	5	D	51,000	Outboard rivet hole at rear-spar end fitting
	6	B	63,000	Outboard rivet hole at end fitting of stiffener 6
	7	B	63,000	Outboard rivet hole at end fitting of stiffener 10
	8	K	84,000	Front spar shear web - fourth outboard lightening hole
	9	G	86,500	Outboard rivet hole at rear-spar end fitting
	10	L	90,500	Rivet hole 2 in. outboard of station 16 next to rear-spar end fitting
	11	D	95,200	Outboard rivet hole at front-spar end fitting
L-5	1	B	5,500	Outboard rivet hole at end fitting of stiffener 5
	2	B	14,500	Outboard rivet hole at end fitting of stiffener 11
	3	H	17,000	Outboard forward corner of large cutout
	4	J	26,000	Rivet hole next to first small cutout
	5	J	30,000	Rivet hole next to first small cutout
	6	B	33,500	Outboard rivet hole in end fitting of stiffener 10
	7	H	34,600	Outboard rearward corner of large cutout
	8	M	40,000	Rivet hole in large cover plate
	9	L	44,000	Rivet hole 2 in. outboard of station 16 next to rear-spar end fitting
	10	I	54,000	Over stiffener 10, 22 in. outboard of station 16
	11	D	73,000	Outboard rivet hole in rear-spar end fitting
	12	D	100,000	Outboard rivet hole in front-spar end fitting
R-5	1	B	5,000	Outboard rivet hole in end fitting of stiffener 5
	2	H	11,000	Outboard forward corner of large cutout
	3	B	22,000	Outboard rivet hole in end fitting of stiffener 6
	4	B	24,000	Outboard rivet hole in end fitting of stiffener 11
	5	H	34,400	Outboard rearward corner of large cutout
	6	J	35,500	Rivet hole next to first small cutout
	7	L	46,000	Rivet hole 2 in. outboard of station 16, next to rear-spar end fitting
	8	I	51,500	Over stiffener 10, 22 in. outboard of station 16
	9	J	51,500	Rivet hole next to second small cutout
	10	J	58,000	Rivet hole next to first small cutout
	11	D	63,000	Outboard rivet hole at rear-spar end fitting
	12	B	71,000	Outboard rivet hole at end fitting of stiffener 10
	13	D	88,000	Outboard rivet hole at front-spar end fitting
	14	G	95,800	Outboard rivet hole at front-spar end fitting
	15	G	95,800	Outboard rivet hole at rear-spar end fitting
L-6	1	B	18,000	Outboard rivet hole at end fitting of stiffener 5
	2	H	21,000	Outboard forward corner of large cutout
	3	B	28,700	Outboard rivet hole at end fitting of stiffener 11
	4	B	39,000	Outboard rivet hole at end fitting of stiffener 6
	5	B	39,000	Outboard rivet hole at end fitting of stiffener 10
	6	H	43,800	Outboard rearward corner of large cutout
	7	D	50,000	Outboard rivet hole at front-spar end fitting
	8	J	51,000	Rivet hole next to first small cutout
	9	L	55,400	Rivet hole 2 in. outboard of station 16, next to rear-spar end fitting
	10	G	66,000	Outboard rivet hole at front-spar end fitting
	11	D	73,500	Outboard rivet hole at rear-spar end fitting
R-6	1	H	11,000	Outboard rearward corner of large cutout
	2	H	25,500	Outboard forward corner of large cutout
	3	A	27,000	Cutout in stiffener 5
	4	J	30,400	Rivet hole next to first small cutout
	5	B	33,000	Outboard rivet hole at end fitting of stiffener 6
	6	B	40,000	Outboard rivet hole at end fitting of stiffener 11
	7	D	54,000	Outboard rivet hole at front-spar end fitting

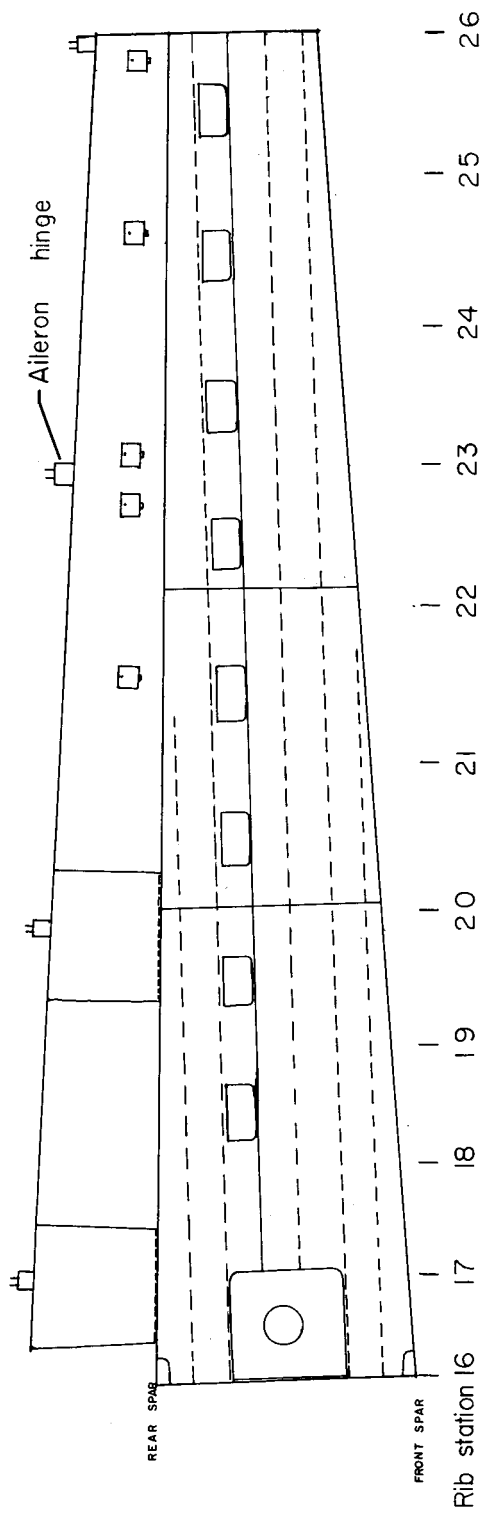
TABLE III
SUMMARY OF CRACK OCCURRENCES

Load level, Δn , g units	Specimen number	Order of occurrence of cracks														Total number of cracks
		Stiffener			Spar			Skin					Shear web	Skin and doubler	Cover- plate	
		A	B	C	D	E	F	G	H	I	J					
												K				
± 0.25	L-7	1	2	4, 6	1			3	5		3		4			6
	R-7		2, 8	6, 7	2			4, 5			1					8
	L-8			3, 6				6								6
	R-8	1	3, 8		5		7	3, 4	2				4			8
	L-9			4, 5	1, 2			3			1					4
R-9					2											5
± 0.42	L-1	1	2		1			2								2
	R-1				4			3								4
	L-2				3		1	2								4
	R-2		2		1, 3			4								4
	L-3	1, 14	3, 7, 12	11	6			13	2, 5	4	8, 9		10			14
R-3	1	3, 4, 8, 10		11, 15			12, 14			2, 5, 6		7	9, 13		15	
± 0.76	L-4	3, 7	5, 6, 8		9, 11			12	1, 2		4				10	12
	R-4	1	2, 3, 6, 7		5, 11			9			4	8	10			11
	L-5		1, 2, 6		11, 12				3, 7	10	4, 5		9	8		12
	R-5		1, 3, 4, 12		11, 13			14, 15	2, 5	8	6, 9, 10		7			15
	L-6		1, 3, 4, 5		7, 11			10	2, 6		8		9			11
	R-6	3	5, 6		7				1, 2		4					7
Total number of occurrences		10	34	10	24	2	2	19	14	3	17	1	8	4		148
Number of times critical		4	19	8	20	2	2	16	8	1	0	0	3	3		



(a) Location.

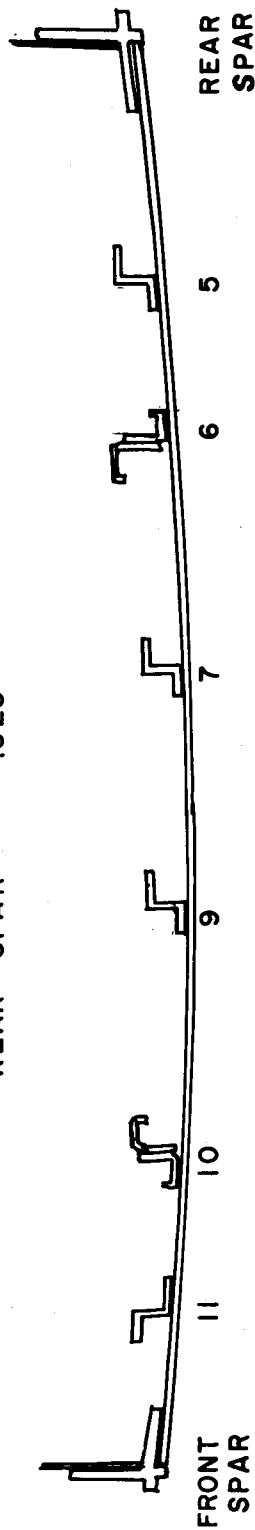
Figure 1.- Test specimen.



(b) Tension surface of test specimen.

Figure 1.- Continued.

FRONT SPAR	=	0.480	SQUARE	INCHES
STIFFENER 11	=	.160	"	"
STIFFENER 10	=	.167	"	"
STIFFENER 9	=	.160	"	"
STIFFENER 7	=	.160	"	"
STIFFENER 6	=	.167	"	"
STIFFENER 5	=	.160	"	"
REAR SPAR	=	.523	"	"



Cross section at rib station 17

(c) Tension surface.

Figure 1.- Concluded.

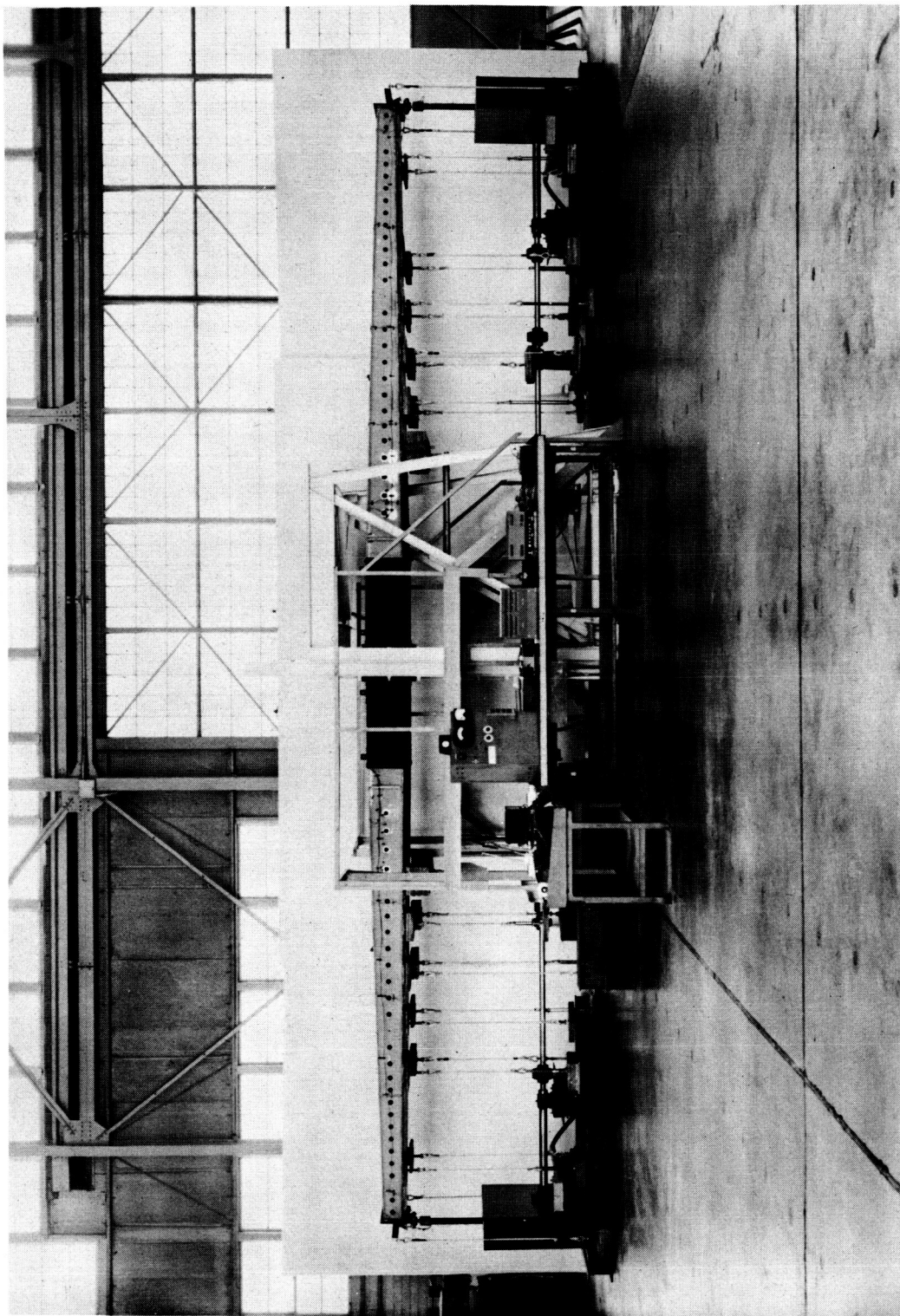


Figure 2.- General arrangement of fatigue machine and specimen. L-57-102

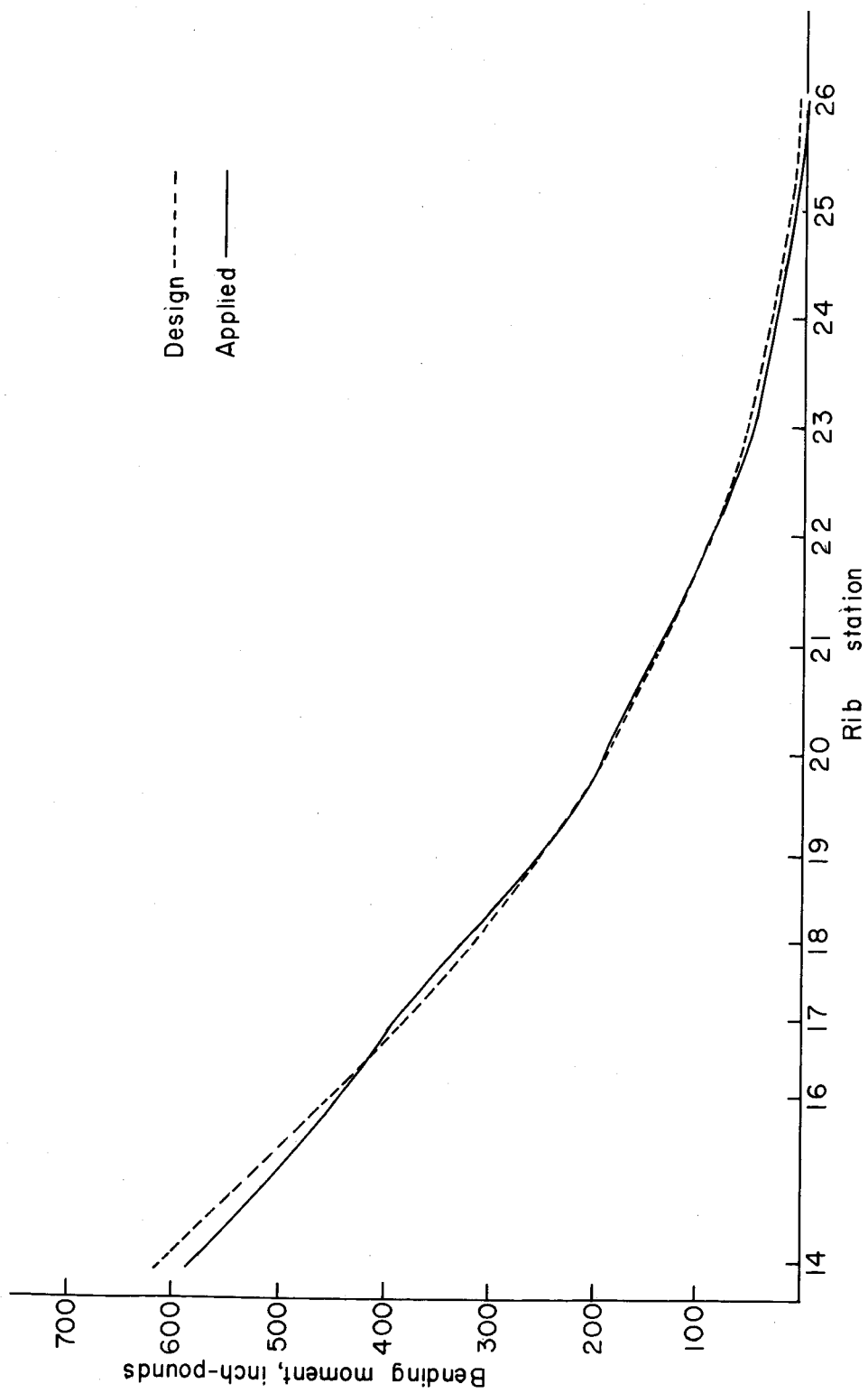
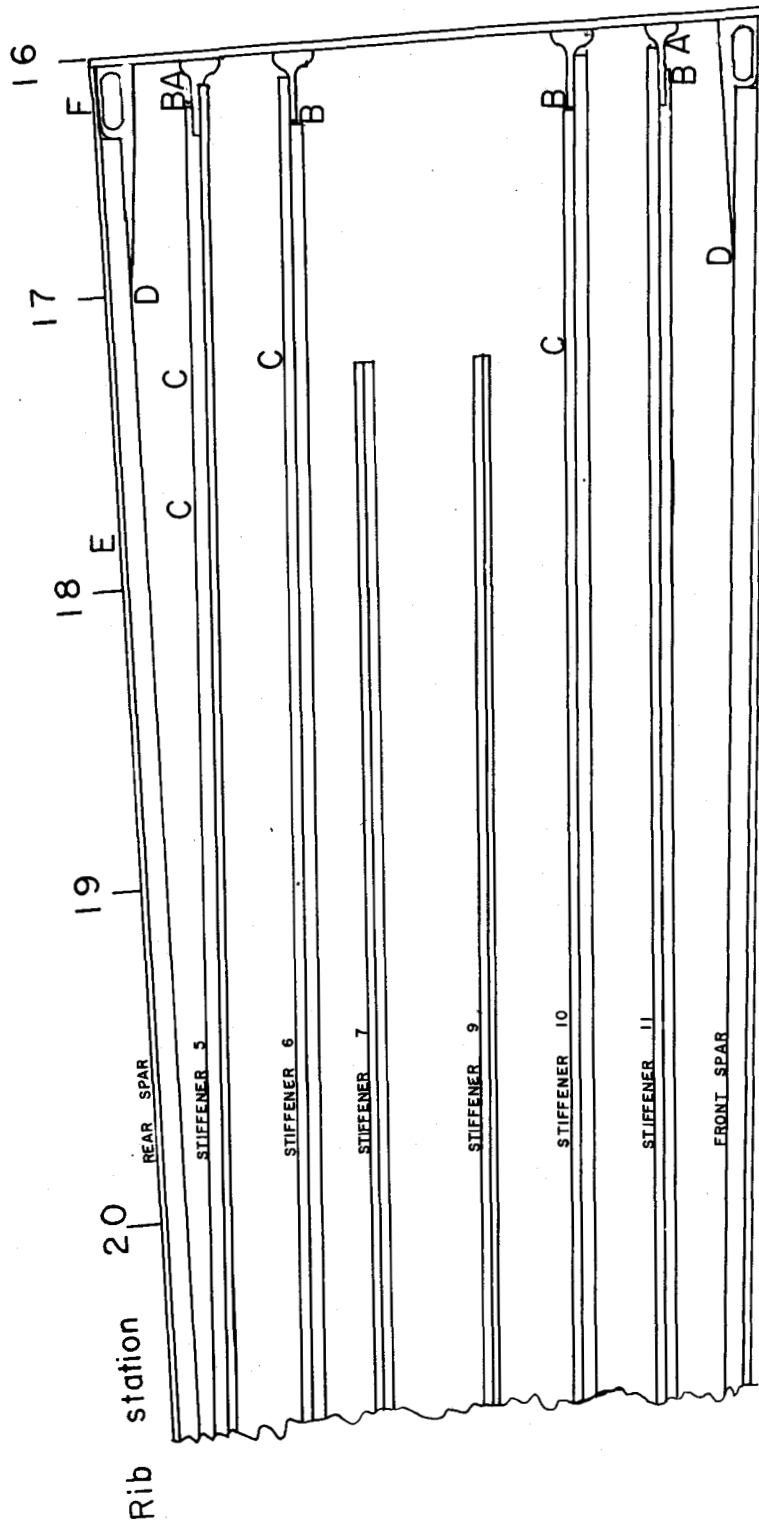
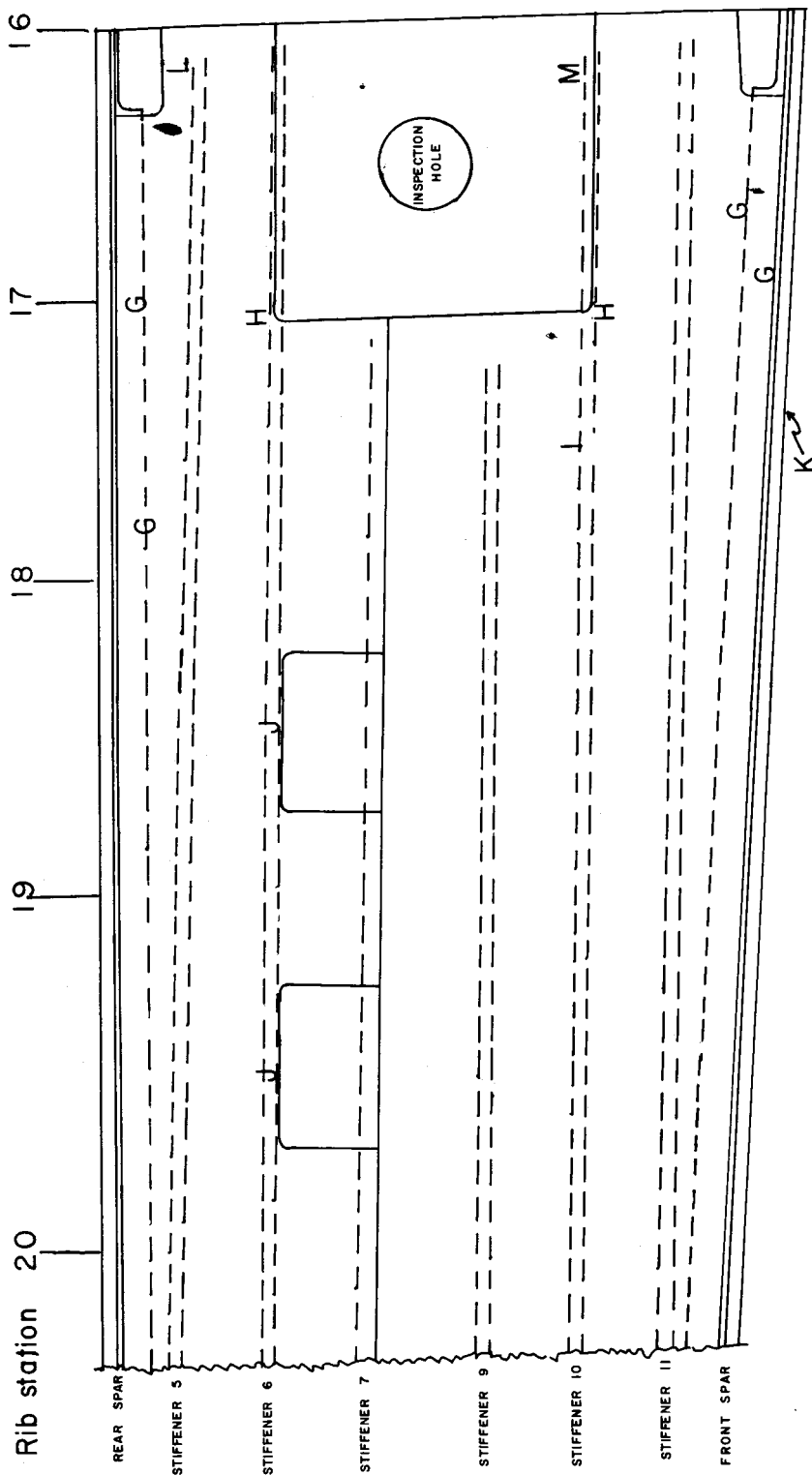


Figure 3.- Bending moment distribution for 1g or level-flight conditions.



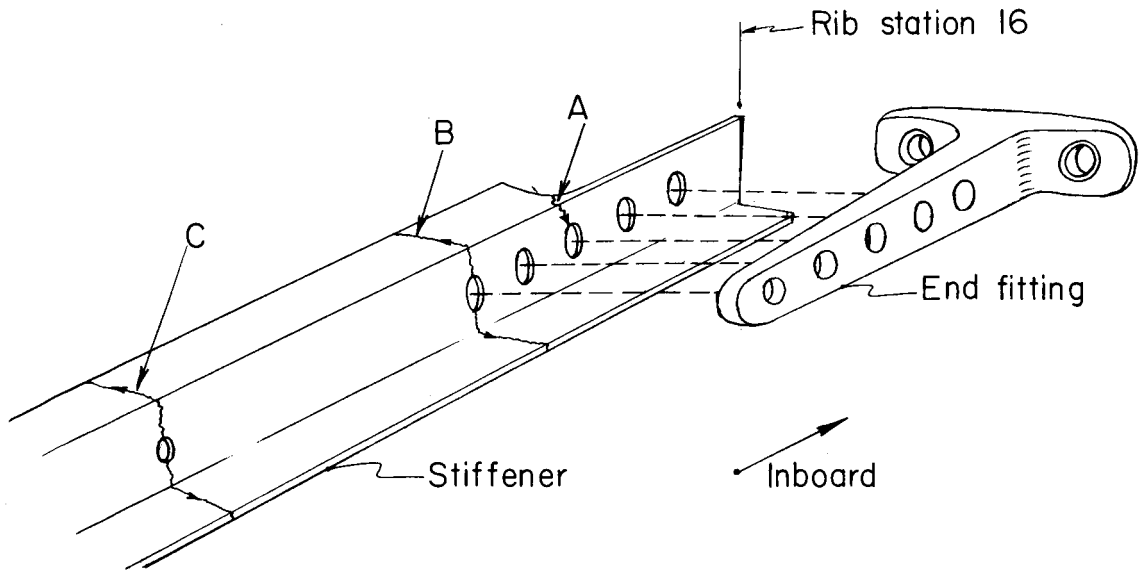
(a) Spar and stiffener crack location.

Figure 4.- General location of cracks.

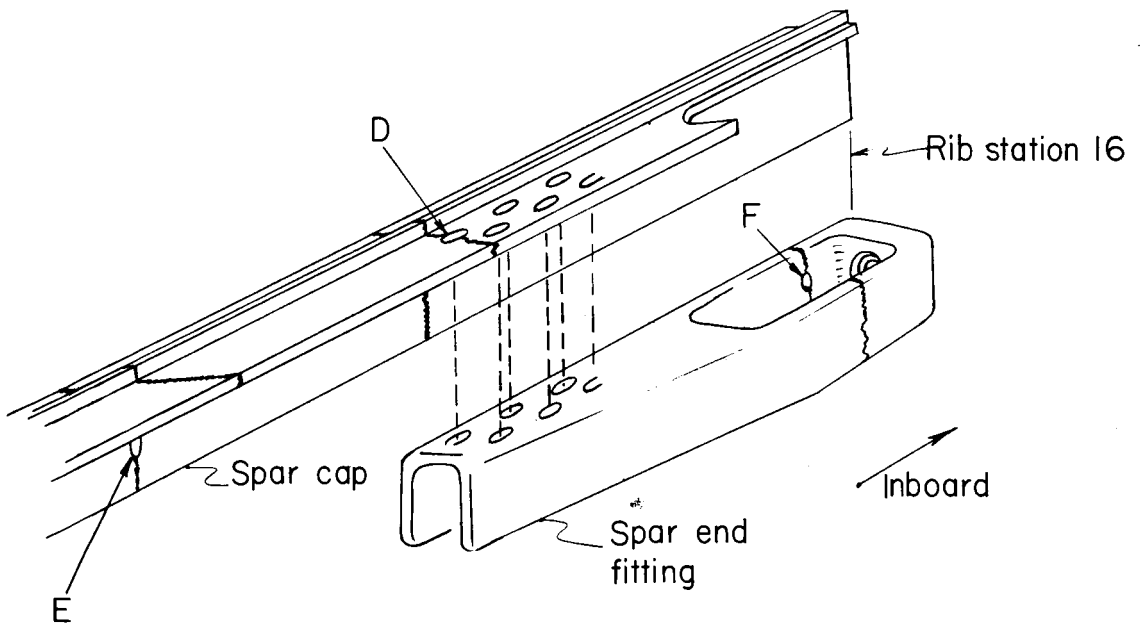


(b) Skin crack location.

Figure 4.- Concluded.

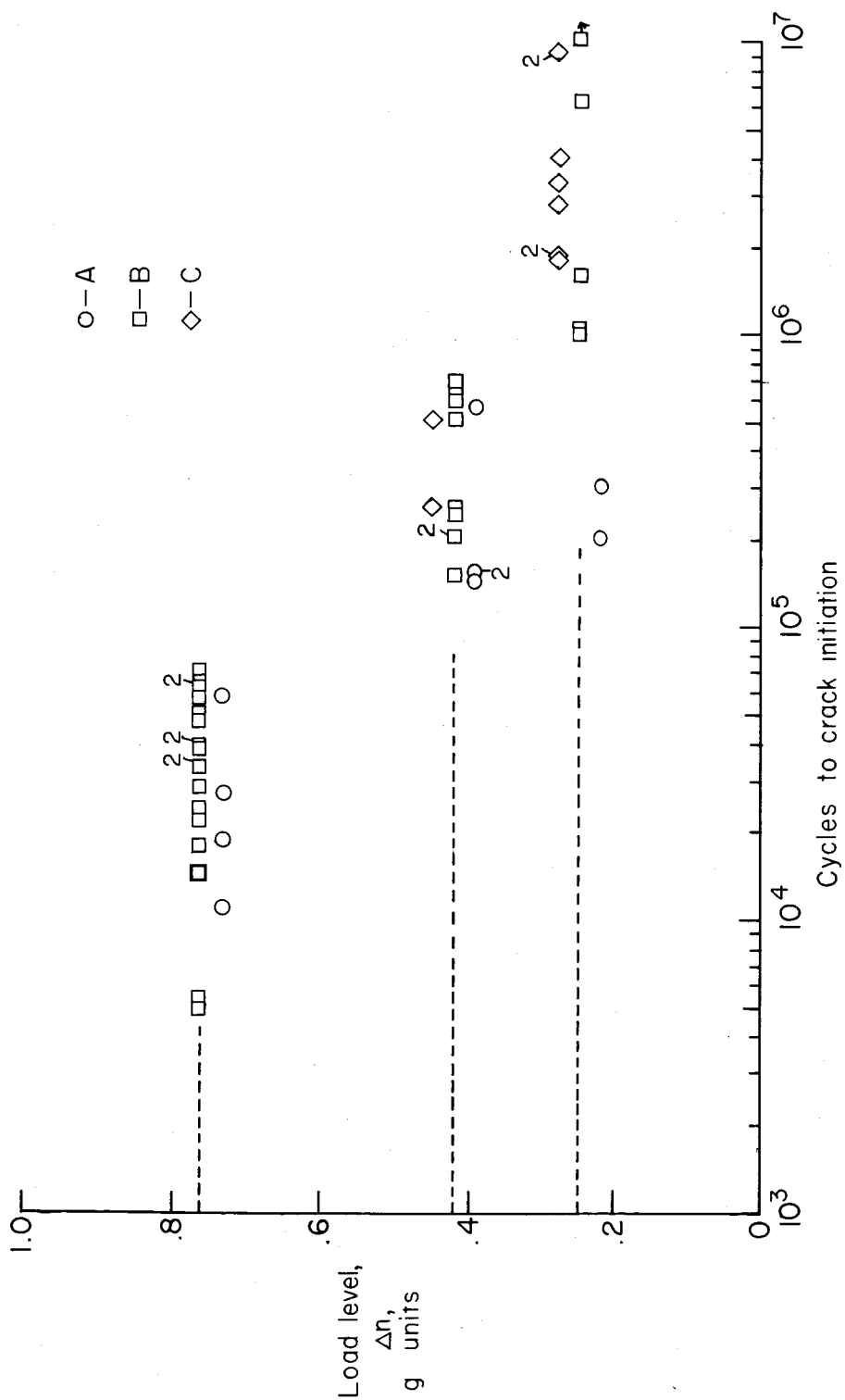


(a) Stiffener cracks.



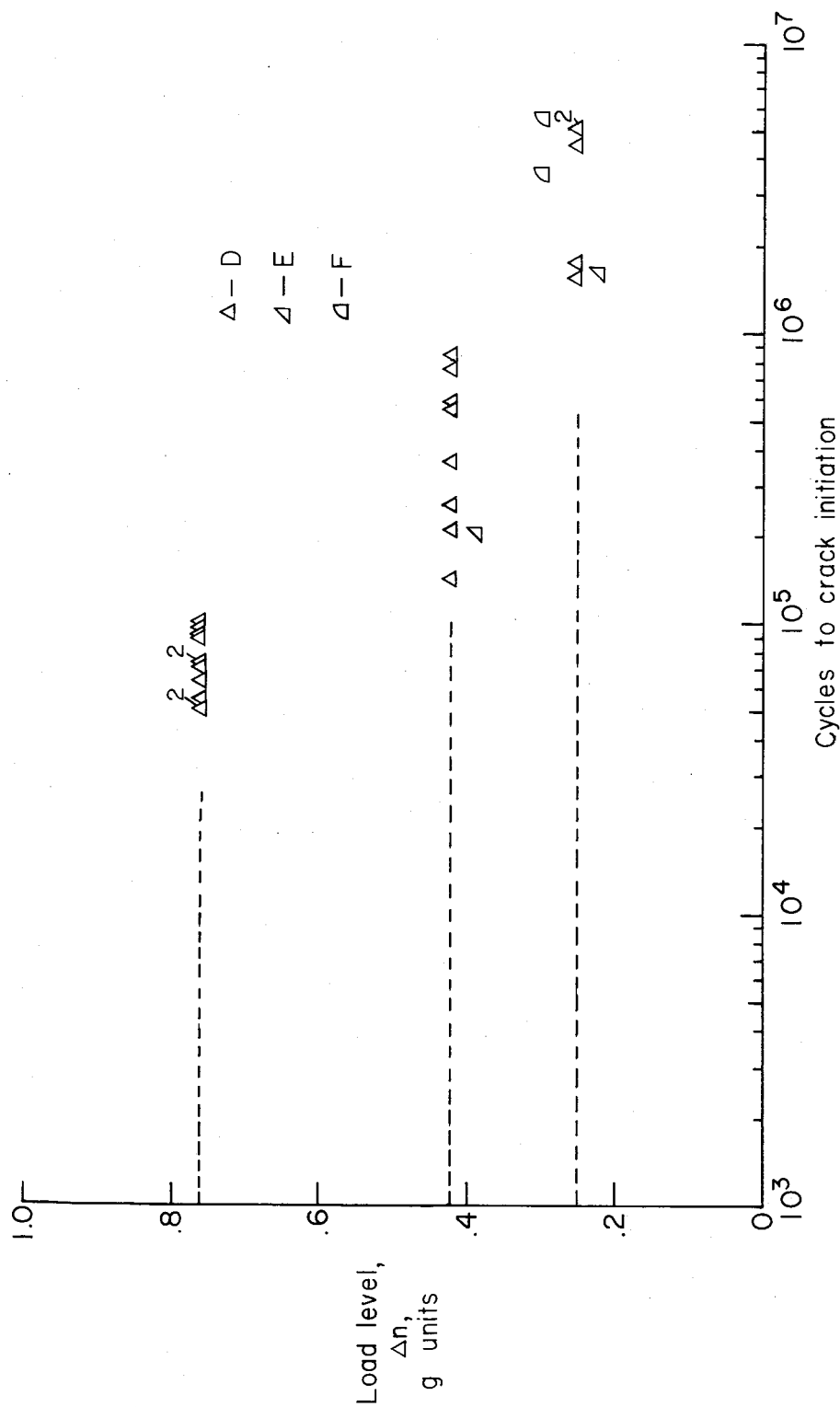
(b) Spar cracks.

Figure 5.- Spar and stiffener crack locations.



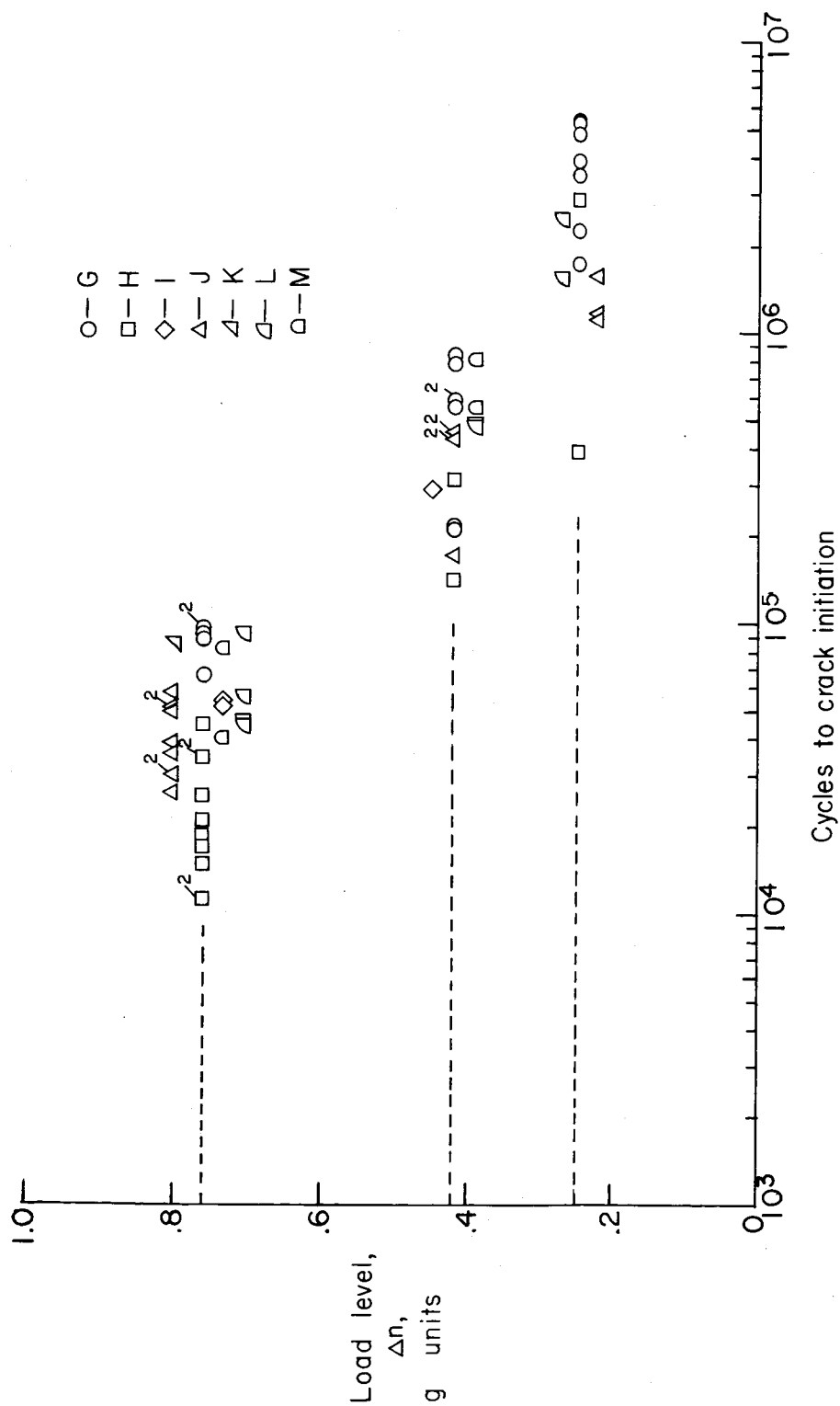
(a) Stiffener.

Figure 6.- Load lifetime curves.



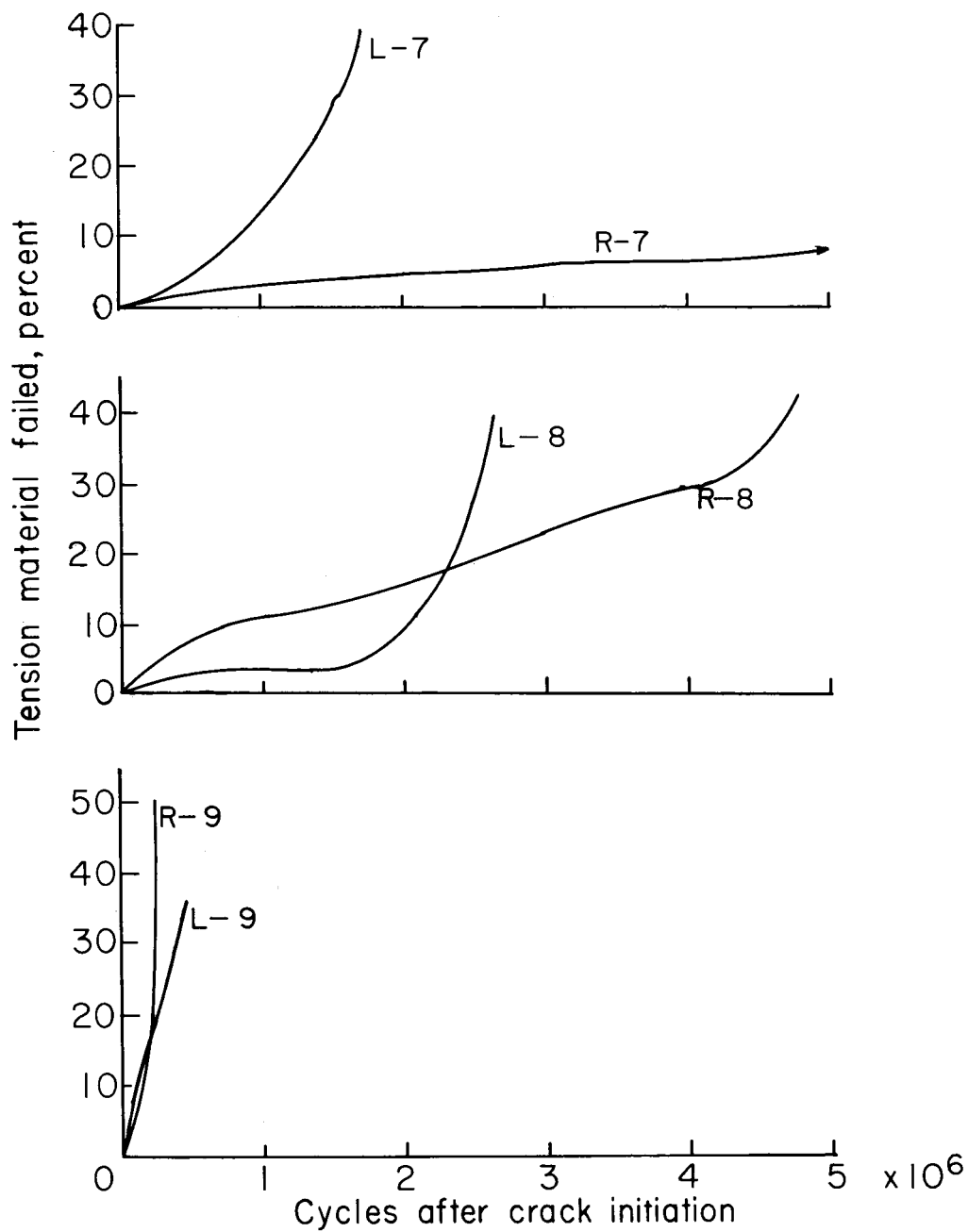
(b) Spar.

Figure 6.- Continued.



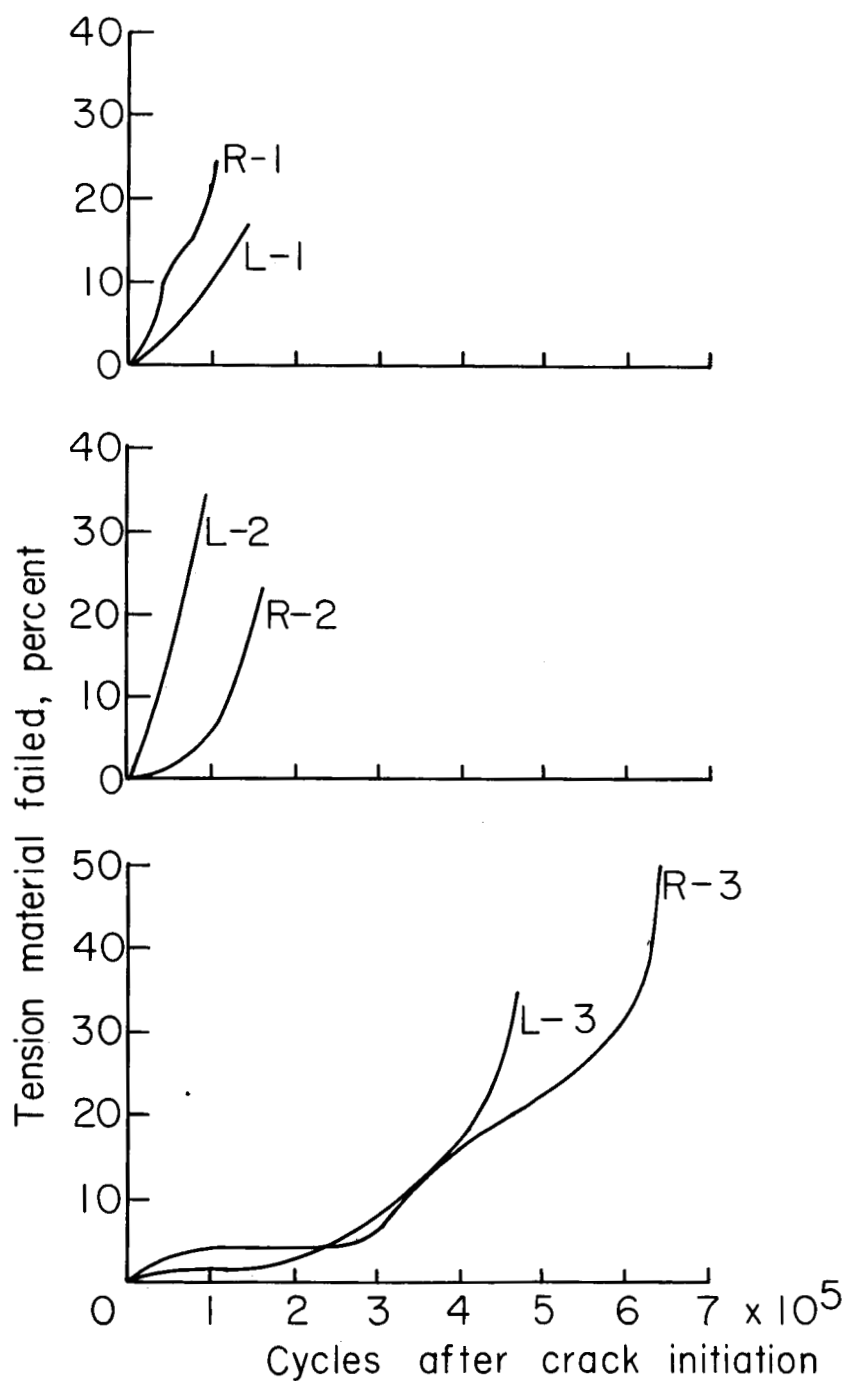
(c) Skin.

Figure 6.- Concluded.



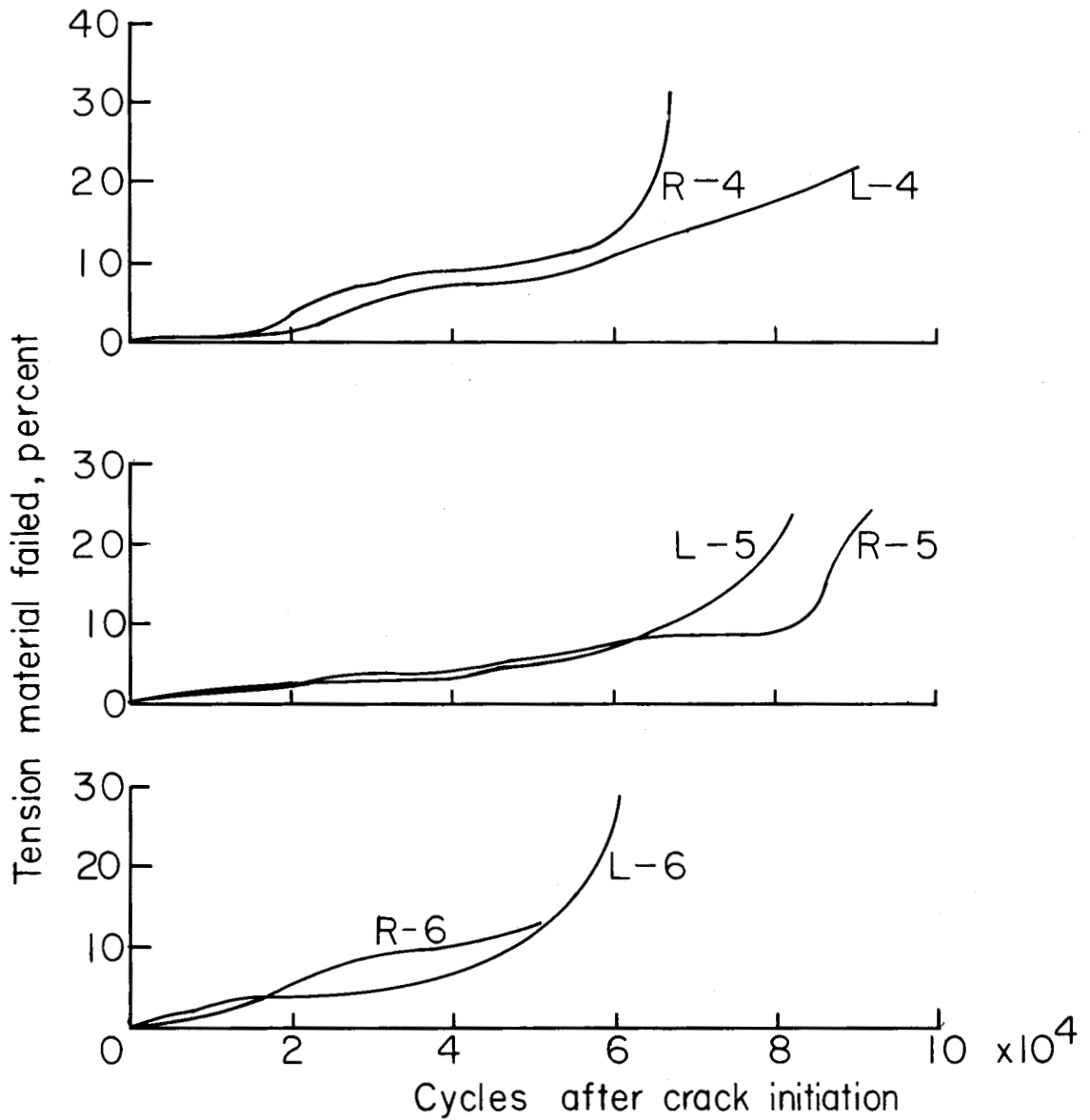
(a) Load level, Δn , $\pm 0.25g$.

Figure 7.- Crack propagation curves.



(b) Load level, Δn , $\pm 0.42g$.

Figure 7.- Continued.



(c) Load level, Δn , $\pm 0.76g$.

Figure 7.- Concluded.

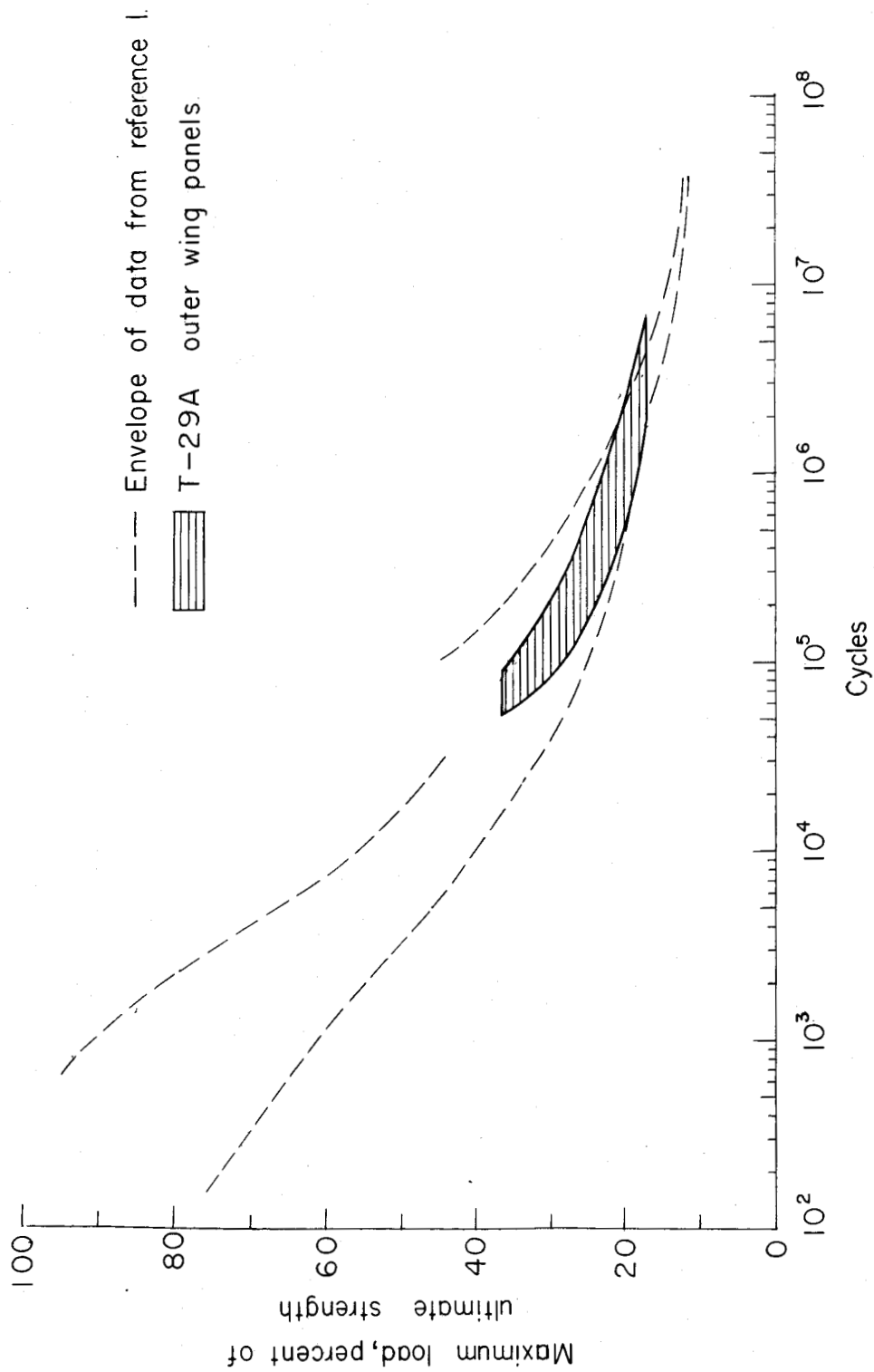


Figure 8.- Load lifetime comparison.

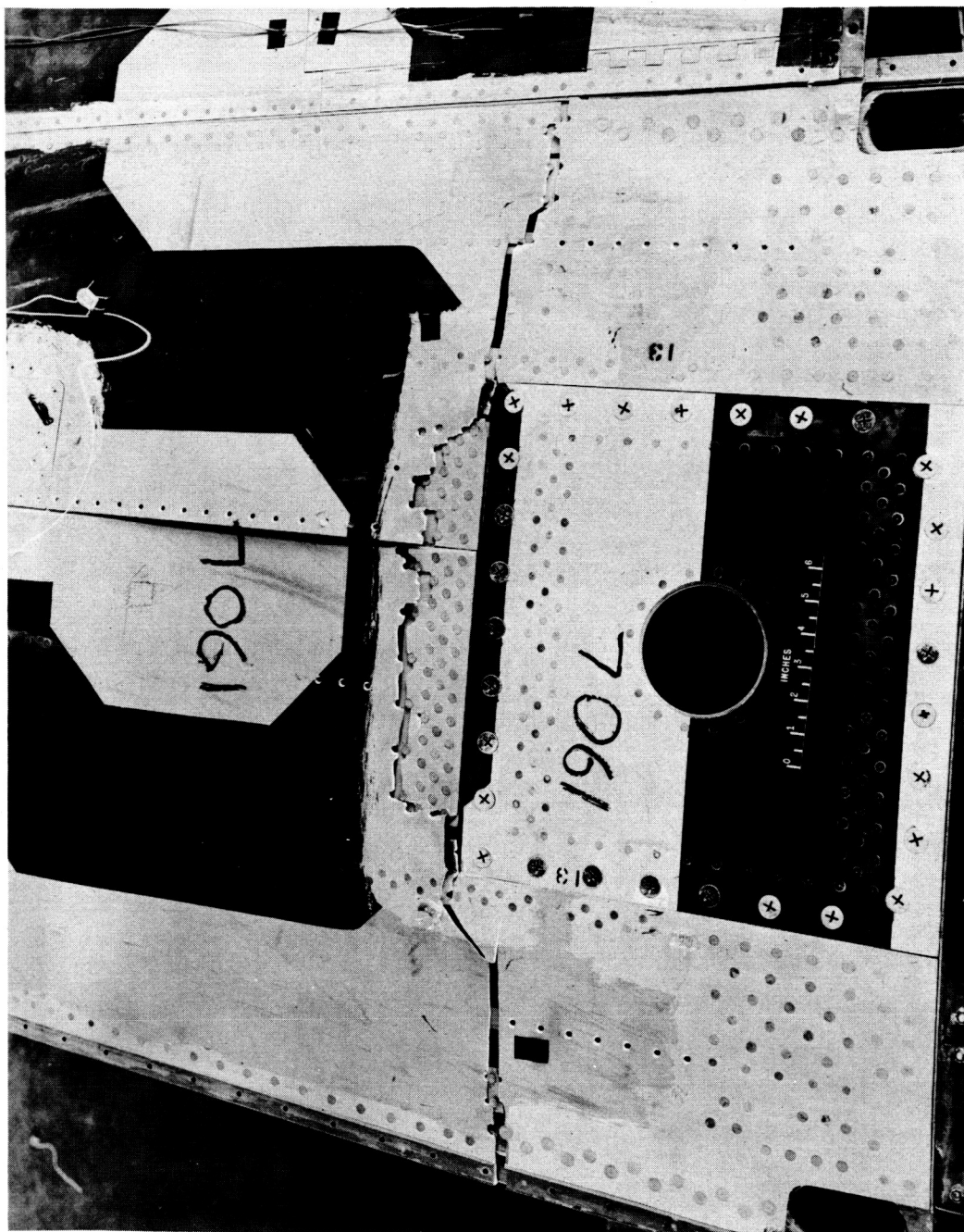


Figure 9.- Test specimen (L-6) following static failure.

L-57-2909

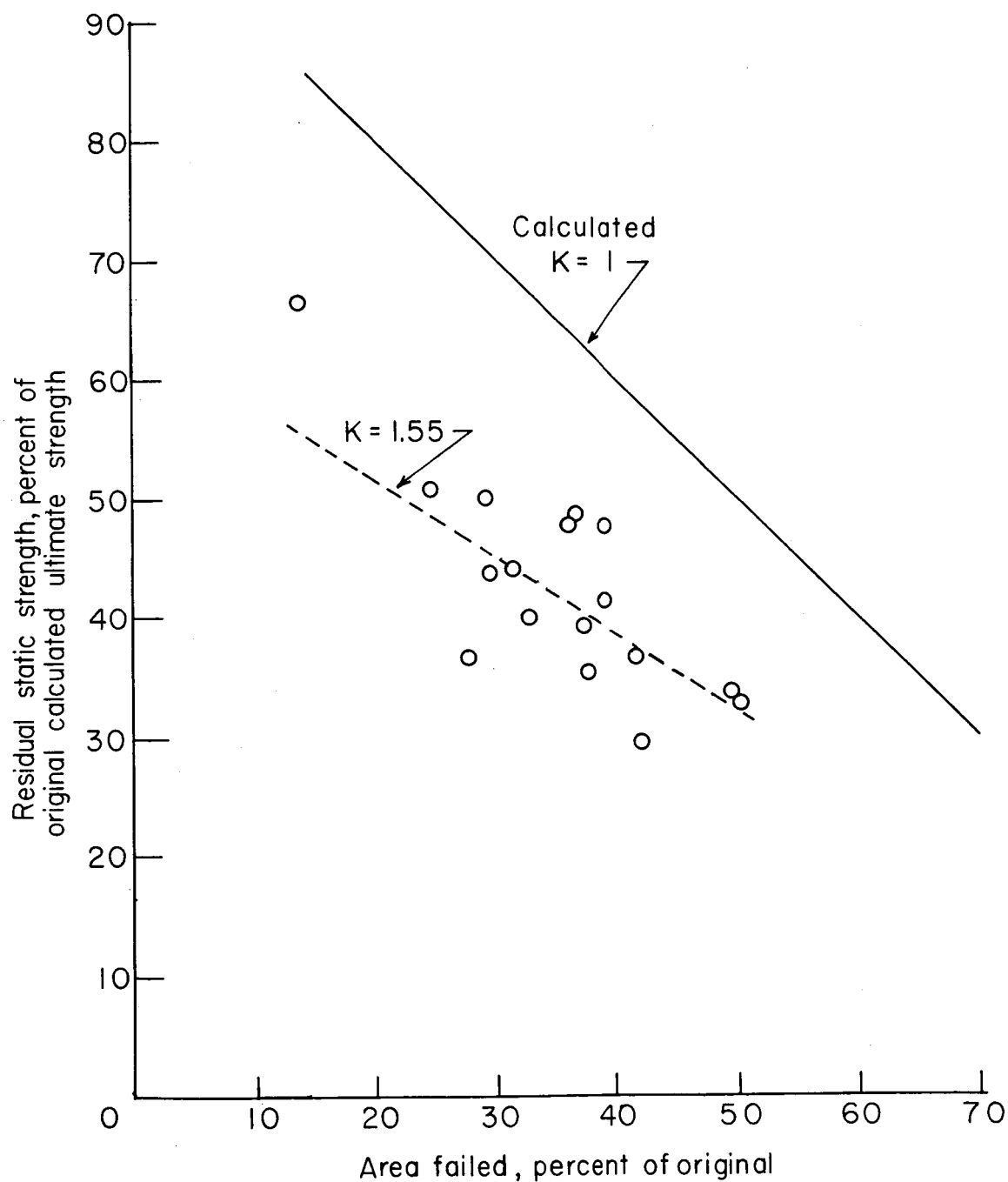


Figure 10.- Residual static strength (K is the static-strength reduction factor).